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WEIGHT TRENDS FOR A FULLY REUSABLE ADVANCED SINGLE-STAGE SHUTTLE

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ERRATA

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WEIGHT TRENDS FOR A FULLY
REUSABLE ADVANCED SINGLE-STAGE SHUTTLE

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September 1983

Page 7, 1st paragraph under Wing Weight Trends (replacement paragraph)

Wing unit weight increases with increased wing size (Figure 12). For example, the wing unit weight for the baseline design would increase approximately 13 percent for a geometrically similar wing subjected to the same wing unit loading but 50 percent larger in area. This trend in unit wing weight was determined from the same methods as those used in reference 8. However the exponential in the geometry term of the wing equation has been increased from 0.572 to 0.879 to better reflect wing unit weight growth with wing size increases. Other equation constants were changed as required to accompany the new exponential. In the previous equation the overall wing weight increased with wing size and loading and reasonably reflected wing weights for wing areas approximating Shuttle wing area. However, wing unit weight (wing weight divided by exposed area) remained essentially constant for wings of equal loading but different areas for the previous equation,whereas unit weight should increase slightly with wing size increases.

Page 22, Figure 12 replace with figure attached.

Issued July 10, 1984

ABSTRACT

The rate at which subsystem weights grow with vehicle gross weight is assessed and is shown to be critical to the efficiency of large Earth-to-orbit transports. Some subsystems grow as the square of vehicle size, others grow as the cube of vehicle size, and still others remain nearly constant irrespective of vehicle size. The overall trend, however, is a reduction in the inerts as a percentage of gross weight as the vehicle size is increased. For this reason, the larger the vehicle, the greater the payload weight delivered per pound of vehicle manufactured.

Other critical issues addressed include the effects of wing loading and wing size on wing weight, the effect of entry planform loading on thermal protection system weight, the impact of power demand on cooling system and prime power weight, and tank fineness ratio on insulation weight. The effects of body shape and various internal packaging arrangements on weight and balance are also discussed. The greatest impact on overall vehicle weight is body shape and internal packaging, and they could account for weight savings of up to 30 percent in body structure. Other subsystems are important, but the savings are much smaller in relation to overall vehicle weight--individually less than one percent.

INTRODUCTION

For every pound of weight added in an Earth-to-orbit transport, approximately 35 pounds of structure, engine, and propellants must be added in order to perform the same mission. (This figure was derived from data presented in Ref. 1 for a dual-fueled single-stage-to-orbit vehicle.) Understanding what factors influence subsystems weights is essential in arriving at the most efficient vehicle design for a given set of mission requirements and design criteria.

Among the factors which impact vehicle weight are payload weight, volume, and shape; the orbital inclination and altitude; the mission duration; and the frequency of missions. Many other factors affect vehicle weight such as services required by the payload including power, communications, and cooling; whether access to the payload is required; and whether extra crew is required to monitor the payload.

Factors related to design criteria which affect vehicle design include how the emphasis is to be apportioned between development, manufacturing, or operational costs. The purpose of this paper is to identify weight trends of some typical subsystems for a circular-bodied single-stage concept. These trends should assist in the design process of future vehicles, especially for those subsystems wherein minimum weight is the primary design goal.

SYMBOLS

K_v tank volume constant

K_w tank weight constant, lb/in^3

ℓ vehicle length, ft

p ullage pressure, lb/in^2

r tank radius, in

S stress, lb/in^2

t tank wall thickness, in

T_i tank weight index, lb/ft^3

V tank volume, ft^3

W_t tank weight, lb

W_p propellant weight, lb

ρ_m mean payload bulk density, lb/ft^3

ρ_p bulk packaged density of LOX/LH_2, lb/ft^3

$\rho_{s/c}$ bulk packaged density of spacecraft, lb/ft^3

λ fineness ratio, barrel length divided by tank diameter

IMPACT OF BODY CONFIGURATION ON WEIGHT

One of the most influential factors governing vehicle weight of Earth-to-orbit transportation systems is body structural weight. Within this subsystem the factors having the greatest influence on weight are external shape and internal packaging. These two factors are nonetheless the most difficult to quantify.

In Ref. 2, a simple circular body was proposed for a single-stage-to-orbit concept with a 65,000 lb payload (Fig. 1). Fuel tanks, payload bay, and oxidizer tank were located in series. Unusable volume was minimized, and the barrel sections of the main tankage served also as the load carrying exterior shell of the vehicle. Superfluous aerodynamic fairings were kept to a minimum and were limited to a nose cone, an intertank adapter section, an aft skirt, and body-to-wing side panels. Because the main propellant tanks were round, pressure-induced loads were limited principally to efficient membrane (tensile) stresses as opposed to noncircular tanks whose walls are subjected to bending from internal pressure. These latter tanks must be internally braced in order to maintain shape, and this adds weight to the structure. Initial concerns in the selection of the circular shape are flyability and heating. From computer flight simulations and preliminary wind-tunnel testing, the configuration appears to be acceptable in both respects.

For a simple similar circular-bodied vehicle (CBV) from Ref. 2, the structure for the series arrangement of tanks and payload was estimated to weigh about one half that of the parallel arrangement of tanks and payload (Fig. 2). The results are based on modeling and structural analysis. The weight difference is due principally to the increased efficiency of volume utilization, multifunctional tank design, and the elimination of fairings and secondary structure.

FACTORS IMPACTING MAIN PROPELLANT TANK WEIGHT

For a tank which is designed by pressure (essentially the case for large tankage on single-stage vehicles), tank wall thickness from the hoop stress formula is:

$$t = \frac{pr}{S} \tag{1}$$

Therefore, for a given tank pressure and a given allowable stress, tank wall thickness is proportional to tank radius. Also, tank weight is given by the product of area times thickness times density of the material of which it is made. For a tank which is enlarged but is geometrically similar, area can be represented by a constant times tank wall thickness times r^2, or tank weight is:

$$W_t = (K_w \times t)r^2 \tag{2}$$

Also, tank volume for geometrically similar but enlarged tankage can be represented by a constant times radius cubed or:

$$V = K_v\, r^3 \tag{3}$$

Substituting Equation (1) in (2) for tank wall thickness, t, tank weight becomes:

$$W_t = (K_w \frac{p}{S})r^3 \tag{4}$$

where the terms within the brackets are all constant for a given material, geometry, and ullage pressure. Now the tank weight per tank unit volume is obtained by dividing Equation (4) by Equation (3) or tank weight index is:

$$T_i = \frac{W_t}{V} = \frac{(K_w\, p/S)\, r^3}{K_v r^3} = \text{a constant} \qquad (5)$$

From the above relationships, the tank weight per unit volume is shown to remain constant for a tank which is designed by internal pressure. As the tank gets bigger, the tank wall gets thicker with no increase in the tank weight index so that the pounds of tank per pound of propellant stored remains constant (Fig. 3). At the same time, as the tank size is increased and the tank wall becomes thicker, joints become easier to make and flaws become smaller compared to the wall thickness. The size of the maximum permissible flaw of 0.050 inches compared to the tank wall thickness is shown in Fig. 4. This wall thickness applies to a forward tank station in the barrel section of the tank. The numerical value of 0.050 inches for flaw size for aluminum was established in reference 3 as inspectable and a maximum size limit. How tank wall thickness affects tank life is illustrated by Fig. 5, a curve repeated from Ref. 4. Also, cryogenic insulation weight required per lb. of stored propellant decreases as tank size increases. This trend is shown in Fig. 6.

Another factor in tank design is fineness ratio. Since the shape with the highest volume-to-surface area ratio is a sphere, the greater the deviation from this shape, the greater the insulation penalty. The trend in insulation weight versus tank fineness ratio ,λ, is shown in Fig. 7 for the LOX tank on the baseline vehicle. The fineness ratio of this tank is 0.94 (tank length for convenience in calculations is taken as the length of the barrel section). Increasing the fineness ratio from 0.94 to three tanks at a fineness ratio of 6.0 would increase insulation weight by an estimated 1,425 lb, a significant weight when considering rocket powered vehicles whose mass fraction is so critical to efficiency.

PAYLOAD BAY ENCLOSURE WEIGHT TRENDS

Like the LOX tank, the payload bay for the baseline vehicle has a low fineness ratio having a 17 ft length by 30 ft diameter, or a $\lambda = 0.57$. The volume is 13 percent greater than for the Shuttle, but it does not have the maximum length capability. The cargo bay wetted area versus λ is shown in Fig. 8. This trend only has significance if the cargo bay has to be enclosed with a shroud or some other type of structure or insulation.

Payload bay design volume is also critical to vehicle final design weight. If most of the cargo bay volume is not needed on most of the missions, then this extra volume represents a structural penalty to be carried to orbit. Projections indicate that over 60 percent of the weight to be

delivered to orbit in the future will be propellants for orbital transfer vehicles. This projection was obtained by estimating the amount of propellant needed on orbit for future mission scenarios (Ref. 5). Most propellants require very little volume compared to other cargo. For example, 65,000 lb of LOX/LH_2 stored in a propellant module with separate tanks at a mixture ratio of 6 to 1 would require only 4062 ft^3. This assumes a stored bulk density of 16 lb/ft^3 when taking into account an allowance for tank end domes, insulation, and a clearance space between LOX and LH_2 tanks.

In addition to propellants, there are other payloads which can also be packaged at high bulk densities. For example, disassembled structure in the form of tapered cylindrical nested columns can be packaged readily at 16 lb/ft^3 (Ref. 6). This means that the percentage of high density payloads may be even higher than the 60 percent quoted when considering delivery of both propellants and dissassembled space structure. As an estimate, using 70 percent for the percentage of cargo at 16 lb/ft^3 (10 percent for space structure and other dense cargo) and 30 percent at a density of 4 lb/ft^3, the composite bulk density is 8.42 lb/ft^3. For example, the NASA Long Duration Exposure Facility has a density of about this latter value (Ref. 7). This facility is an open circular frame 14 ft in diameter to which experiments can be attached, but its bulk density is approximately 4 lb/ft^3 (fairly typical for spacecraft). The mean bulk density of cargo for the overall life of the vehicle can be calculated as follows:

$$\rho_m = \frac{W_p + W_{s/c}}{\dfrac{W_p}{\rho_p} + \dfrac{W_{s/c}}{\rho_{s/c}}} \qquad (6)$$

For the above assumptions for payload mix and bulk densities, the mean bulk density is 8.42 lb/ft^3. Based on this average and a 65,000-lb payload capability, the required payload bay volume would only be 7720 ft^3 for both spacecraft and propellants at a 70/30 mass split (Point 1 in Fig. 9). If, however, the volume were to be filled entirely with propellant at 16 lb/ft^3, the required payload volume would be 4062 ft^3 (Point 2). On the other hand, if the cargo bay were to be filled with spacecraft, the required volume would be 16,250 ft^3 (Point 3 in Fig. 9). The penalty for unused cargo bay volume for the baseline design is estimated at 730 lb per 1000 ft^3 of volume. This is based on an estimate for weight penalty increases in the intertank adapter length to provide the added volume. The penalty would be much greater for high fineness ratio payload bay cavities because of higher payload bay wetted areas and the attendant impact on the overall vehicle size.

Mixed manifesting, that is, combining propellant delivery and other cargo on the same flight to maximize the use of the vehicle volume as well as the maximum lift capability, may be the solution (Point 1 in Fig. 9). A staging area on orbit, such as space station, would facilitate such an approach whereby propellant and spacecraft could be collected in partial deliveries to be assembled later to satisfy some future mission.

IMPACT OF CREW AND PAYLOAD LOCATION ON WEIGHT

Location of the crew module and payload in the vehicle has a substantial impact on overall vehicle weight and balance. On any air transport, the crew compartment (or flight deck) is located near the nose, and the payload is distributed aft in such a manner as to minimize the c.g. excursion between the full and empty payload conditions. Thus, trim losses are minimized for the various loading conditions.

From the standpoint of pilot visibility and vehicle weight and balance, the appropriate location for the crew is near the nose of the vehicle. For a smaller vehicle, the entry design c.g. can be improved (moved forward approximately 3 percent of reference body length) with the crew in the nose instead of in the intertank section (comparisons 1 and 2 in Fig. 10). As the vehicle becomes bigger, the effect is still significant but is less pronounced (comparison 3 in Fig. 10).

Because these vehicles tend to have aft c.g.'s, the forward location of the crew module is an advantage. The adverse effects of a rearward c.g. location were assessed in Ref. 9 wherein it was shown that excessively large downward deflections of the body flap were needed in order to trim the vehicle hypersonically. This excessive downward deflection causes excessive heating of the body flap and increases the thermal protection system weight on the deflected control surface.

The c.g. excursions for different entry payload loadings also impact vehicle weight in that greater actuator power is needed for the aerodynamic control surfaces and (like the aft c.g. case just cited) more thermal protection is needed on control surfaces for the rearward c.g. entry cases. In order to minimize vehicle entry c.g. excursions for payload-in versus payload-out cases, the logical location for the payload bay is in the vicinity of the nominal empty vehicle c.g. (station 3 in Fig. 11).

Unfortunately, the location of the crew module near the nose and the payload bay aft is not compatible with most space operations wherein crew module and payload bay (such as the Shuttle) are adjacent so that mission specialists can easily observe activities in the payload bay. Also (with the two adjacent) power, communications, and environmental controls systems, common to both crew module and payloads, can be shared. The excursions of the vehicle entry c.g. with changes in payload weight and location can be seen in Fig. 11 for payload located in the nose, in the intertank section, and aft at the nominal vehicle c.g.

For comparison, a 2 1/2 percent c.g. excursion is shown which is the variance allowed in the Shuttle design. The top (horizontal line) in Fig. 11 depicts a hypothetical payload location always at the nominal entry c.g. (0.72ℓ). If the payload is small in physical size and mass, a payload bay near the nose of the vehicle may be acceptable. For example, a 13,000 lb payload in the nose gives a c.g. within the c.g. excursion limit (Point 1 in Fig. 11). This vehicle, designed for a 65,000 lb payload, has an entry weight of 303,154 lbs with the 13,000 lb payload. Vehicles designed for delivery of much smaller payloads would be lighter and, of course, more sensitive to payload-in versus payload-out cases.

Although the aft payload location yields minimum c.g. excursion (ideally zero), the location does not coincide with the intertank space when the optimal splits between fuels and oxidizer are used. Because of the vehicle balance and crew-payload access problems, a compromise location was selected for both the crew compartment and payload bay, namely in the intertank section even though the pilot's visibility is limited.

WING WEIGHT TRENDS

Wing unit weight decreases slightly with increased wing size (Fig. 12). For example, the wing unit weight for the baseline design would be reduced 1.4 percent for a geometrically similar wing subjected to the same wing unit loading but doubled in area. This trend in unit wing weight was determined from methods used in Ref. 8 for calculating wing weights for various wing geometries.

See errata

Using the same methods, wing unit weights were calculated for various types of wings and various wing loadings (Fig. 13). The external geometry of the wings shown is similar, but the type and materials of construction differ. The uppermost curve in the figure applies to a wing designed to carry propellants and is of aluminum skin-stringer construction. There are very few dry wing aircraft in existence today to compare with because of the necessity for placing the fuel in the wings for purposes of load relief for horizontal takeoff. This design practice is not necessary for vertical takeoff Earth-to-orbit transports, preliminary estimates suggesting that storing propellant in the wings with the required cryogenic insulation would result in a heavier vehicle than providing the equivalent tankage in the vehicle body. From knowledge of current aircraft, it is estimated that a cryogenic wet wing would be 20 to 30 percent heavier than a dry wing. (Compare upper two curves in Fig. 13.)

If an aluminum skin-stringer dry wing is replaced by a wing with composite construction, an additional 25 percent savings in weight is projected (third curve from top in Fig. 13). If the composite skin-stringer dry wing is replaced by a wing of advanced structural design such as that described in Ref. 10, an additional weight reduction is projected (bottom curve in Fig. 13). Another trend illustrated by the figure is that of wing weight versus wing loading. For example, doubling wing loading from 40 to 80 lb/ft^2 increases wing weight by 32 percent - a substantial amount, but not linear with load.

The low wing weight projected for this Earth-to-orbit transport is the result of several factors other than materials selection and details of structural design just cited. Example factors include much thicker sections than any other clipped delta wing having chord thickness-to-length ratios of 10 to 12 percent versus 3 to 6 percent for most military aircraft; much lower limit design maneuver loads than most aircraft (2-1/2 g's versus 5 to 7 for most aircraft); absence of propellant tankage in the wing; absence of main landing gear compartments (in most cases) since the gear can be stowed more conveniently in the much wider bodies of these vehicles; and as a final example, fewer aero-control surfaces such as spoilers and high-lift devices.

THERMAL PROTECTION SYSTEM WEIGHT TRENDS

As geometrically similar Earth-to-orbit transports are altered in size, entry planform loading changes (Fig. 14). For very small vehicle sizes, fixed weights become dominant. These fixed weights include navigation and communication aids, crew and crew support, and crew module. As the vehicle grows, the fixed weights become an increasingly smaller fraction of the variable weights while propellant tanks and main rocket engines increase roughly as vehicle length cubed. Other elements such as wings and fairings increase only as the exponential of two. These trends affect vehicle density and, therefore, the vehicle planform loading at entry. (Mass distribution is also affected as evidenced by the c.g. movement rearward shown in Fig. 10 with vehicle weight increase.)

To give some idea of the effect of design return payload, for example, on the thermal protection system design unit weight, consider the baseline vehicle. If the design entry payload were to be limited to 32,000 lb (the Shuttle limit) instead of the 48,000 lb allowable based on c.g. constraints, then the thermal protection system (TPS) unit weight could be reduced from 1.55 to 1.49 lb/ft^2 or by an estimated 1272 lb or 3.4 percent (compare Points 1 and 2 in Fig. 15). These curves are based on methods used in Ref. 8 for determining TPS average unit weights.

MISCELLANEOUS SUBSYSTEM WEIGHT TRENDS

Many other subsystem weight trends are important. In Fig. 16, the ratio of fixed weight to useful payload weight is shown versus vehicle gross weight. For the baseline vehicle, this value is six tenths of a pound of navigation and life support equipment and crew which must be carried to orbit for every pound of payload. For a vehicle weight of 2M lb gross, this figure has increased to 6 lb of fixed weight for every pound of useful payload.

Increasing on-orbit staytime and number of crew adds significantly to vehicle weight. For example, carrying a crew of six for 14 days requires a weight in personnel provisions (food and clothing, etc.) of 4100 lb, whereas two crewmen for 2 days only requires 1900 lb in personnel provisions (data from Ref. 11 plotted in Fig. 17).

Another factor impacting vehicle weight is power demand. For every killowatt-hour required by the vehicle's avionics, the prime power and environmental control systems must be increased in size and weight. Not only do the electronics have to be cooled as a result of the added power supplied to them, but the fuel cells themselves require cooling since approximately 45 percent of the reactant energy produced by the LOX and LH_2 is given off as heat. The weight of the prime power supply and its reactants and the weight of the cooling system required per kilowatt used by the avionics are shown in Fig. 18. The fuel cells and the cooling system dry weights are basic to the system, but for each increment of mission time more fuel-cell reactants (and their dewars) and cooling system expendables must be added. Not only is more cooling system needed for each kilowatt of power demand, but the cooling system itself requires power to operate fans, pumps, and controls. Therefore, both the absolute weight of the individual avionics subsystem and its power requirements are important. These estimates were derived from a Rockwell detailed weight statement (Ref. 12).

CONCLUSIONS

The following are the major conclusions from a study of the subsystem weight trends for Earth-to-orbit transports:

1) Placing the crew module in the nose of the vehicle improves vehicle overall c.g. and pilot visibility, but this location does not facilitate access to those payloads which must be placed more rearward in the vehicle nearer the nominal c.g. for reasons of balance.

2) Payload bay (and crew module) for smaller payloads can be located in the forebody near the nose of the vehicle, but the payload must be limited in size and weight because of geometric constraints and vehicle balance (the latter restraint applying to entry).

3) Packaging the vehicle with payload and main propellant tankage in series, in a circular shell with crew and payload in the intertank section, is lighter than a parallel arrangement.

4) As the Earth-to-orbit transport is increased in size, several beneficial weight trends result, namely: the main propellant tank walls become thicker (based on pressure design) with no accompanying penalty in tank weight fraction; secondly, cryogenic insulation weight fraction decreases since propellant volume is increasing by dimension cubed and insulation area is increasing only by dimension squared; thirdly, the fixed weights such as crew and avionics become an even smaller percentage of vehicle weight.

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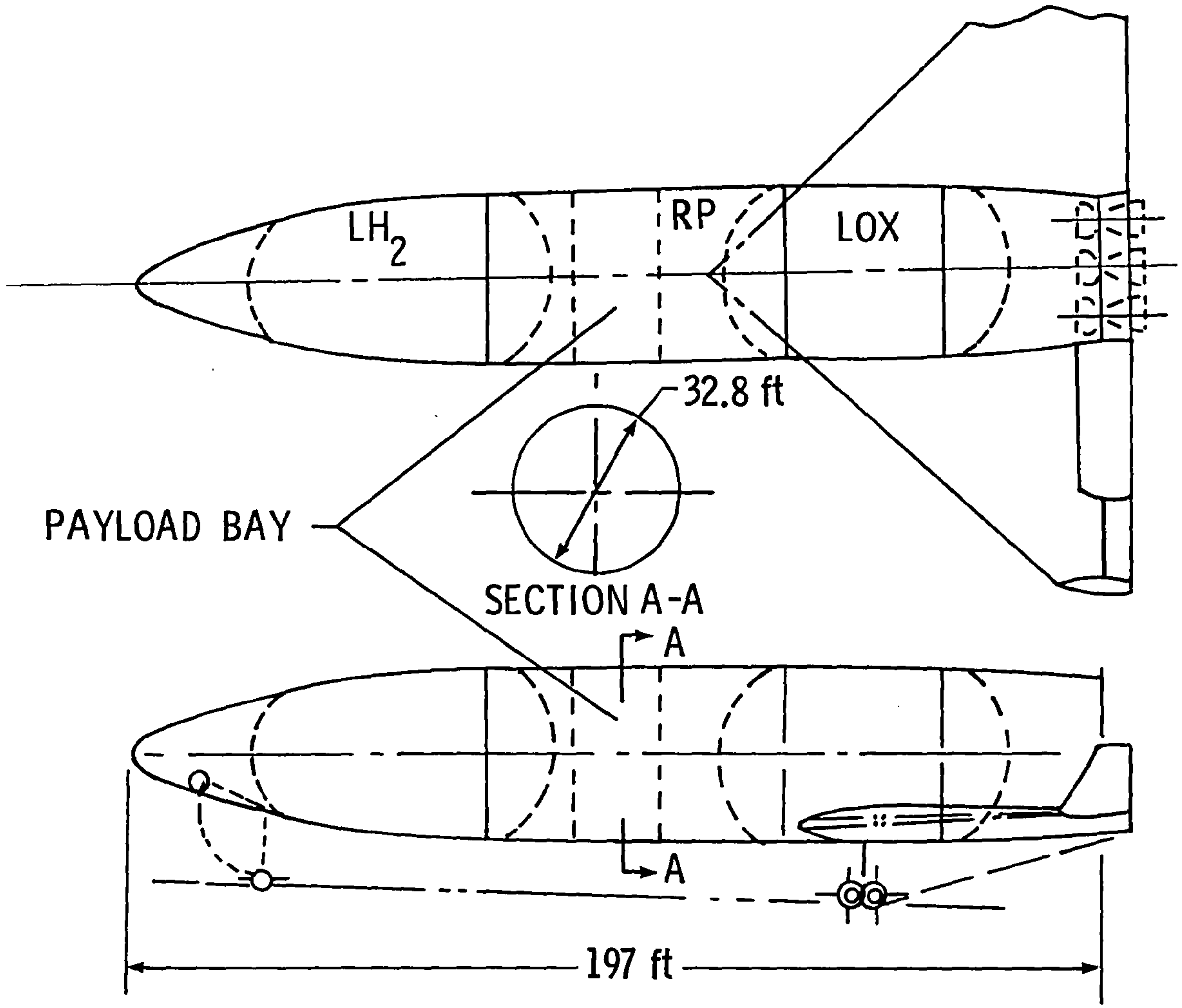

FIG. 1 Baseline vehicle. A single-stage-to-orbit concept with symmetrical forebody (65000 lb payload capability).

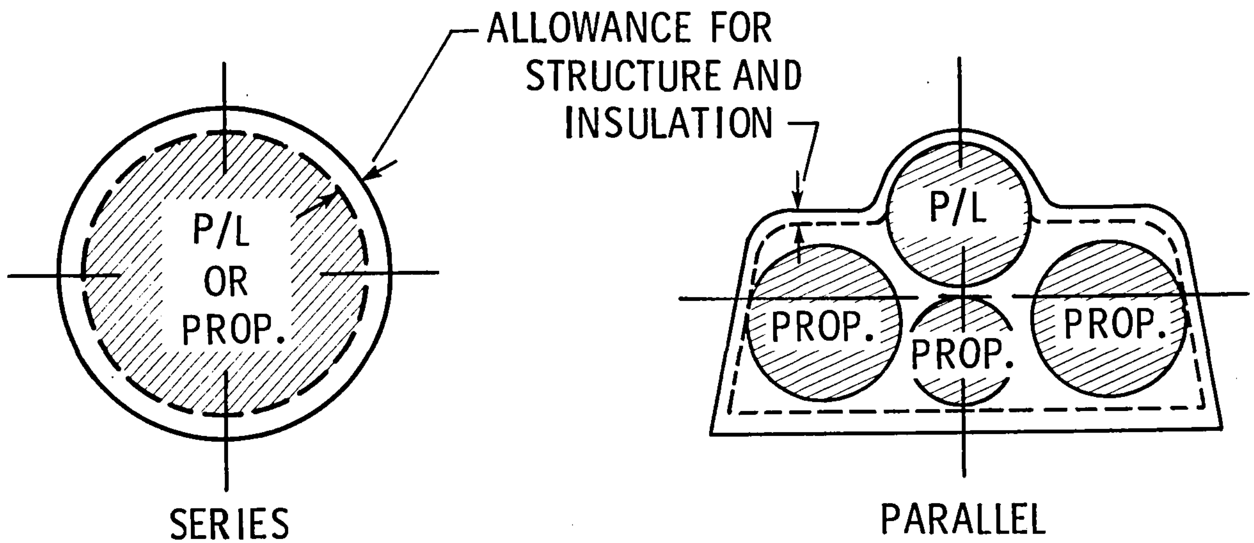

FIG. 2 Comparison of series and parallel packaging of payload and propellants.

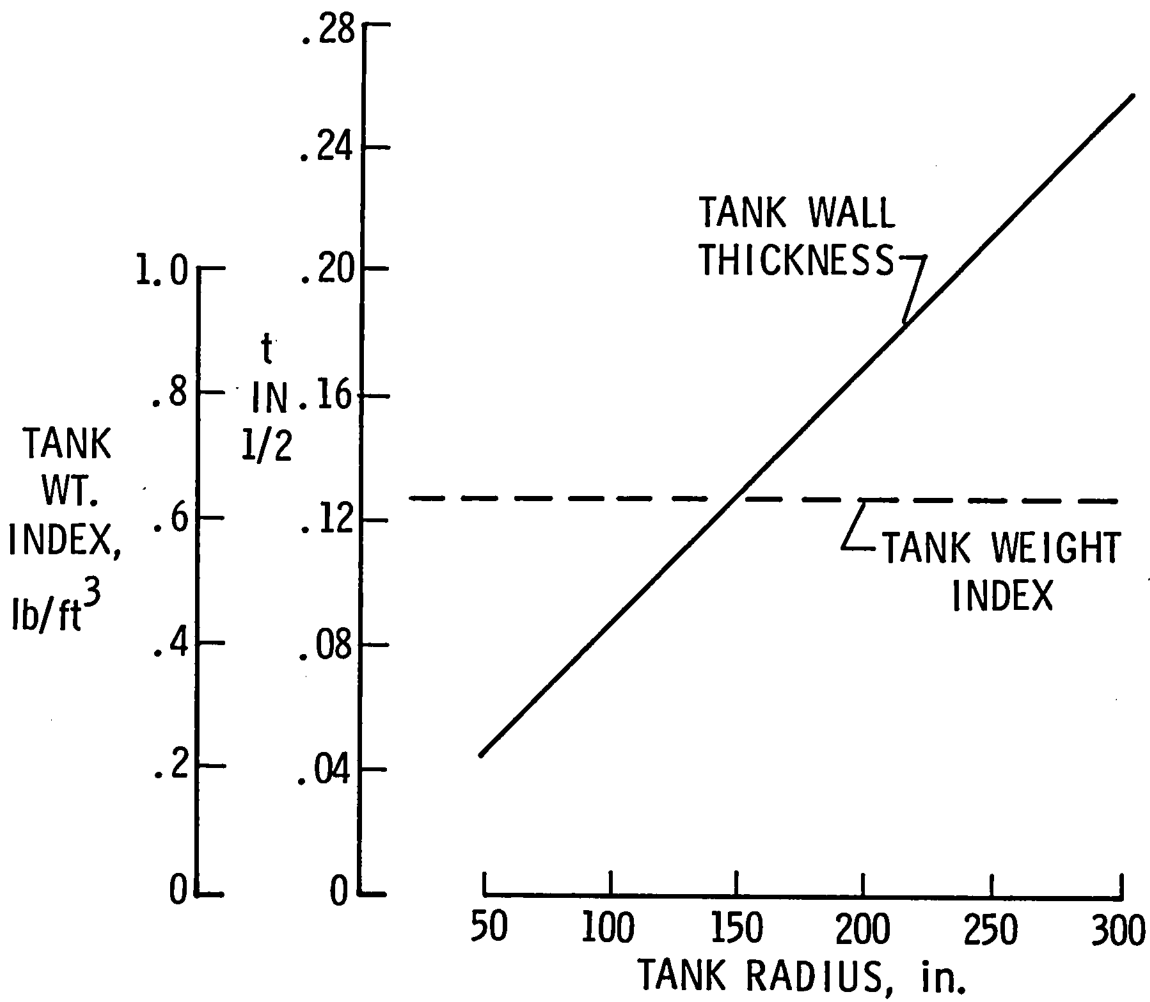

FIG. 3 Tank weight index and tank wall thickness versus tank radius for a circular crossection tank.

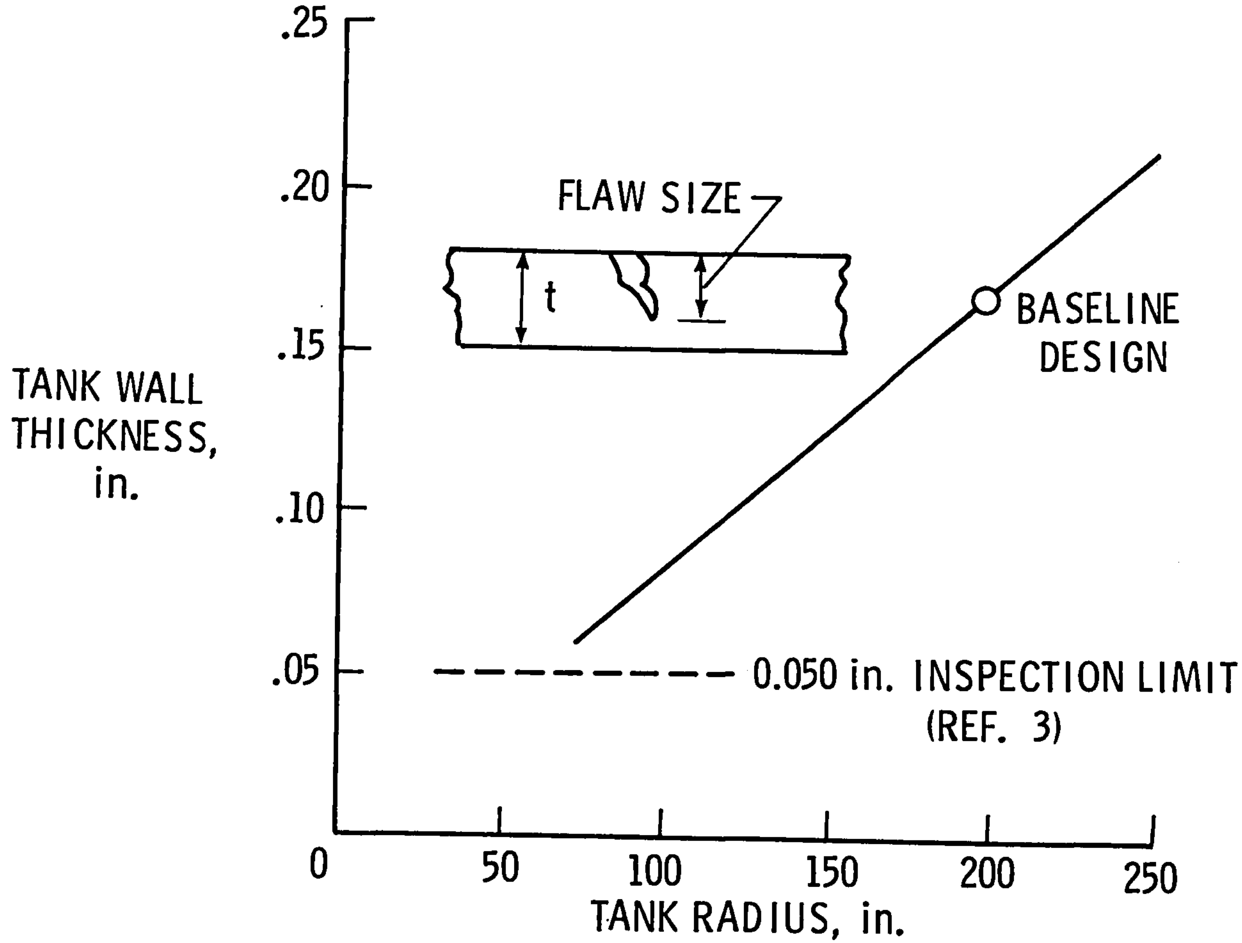

FIG. 4 Tank wall thickness versus tank radius.

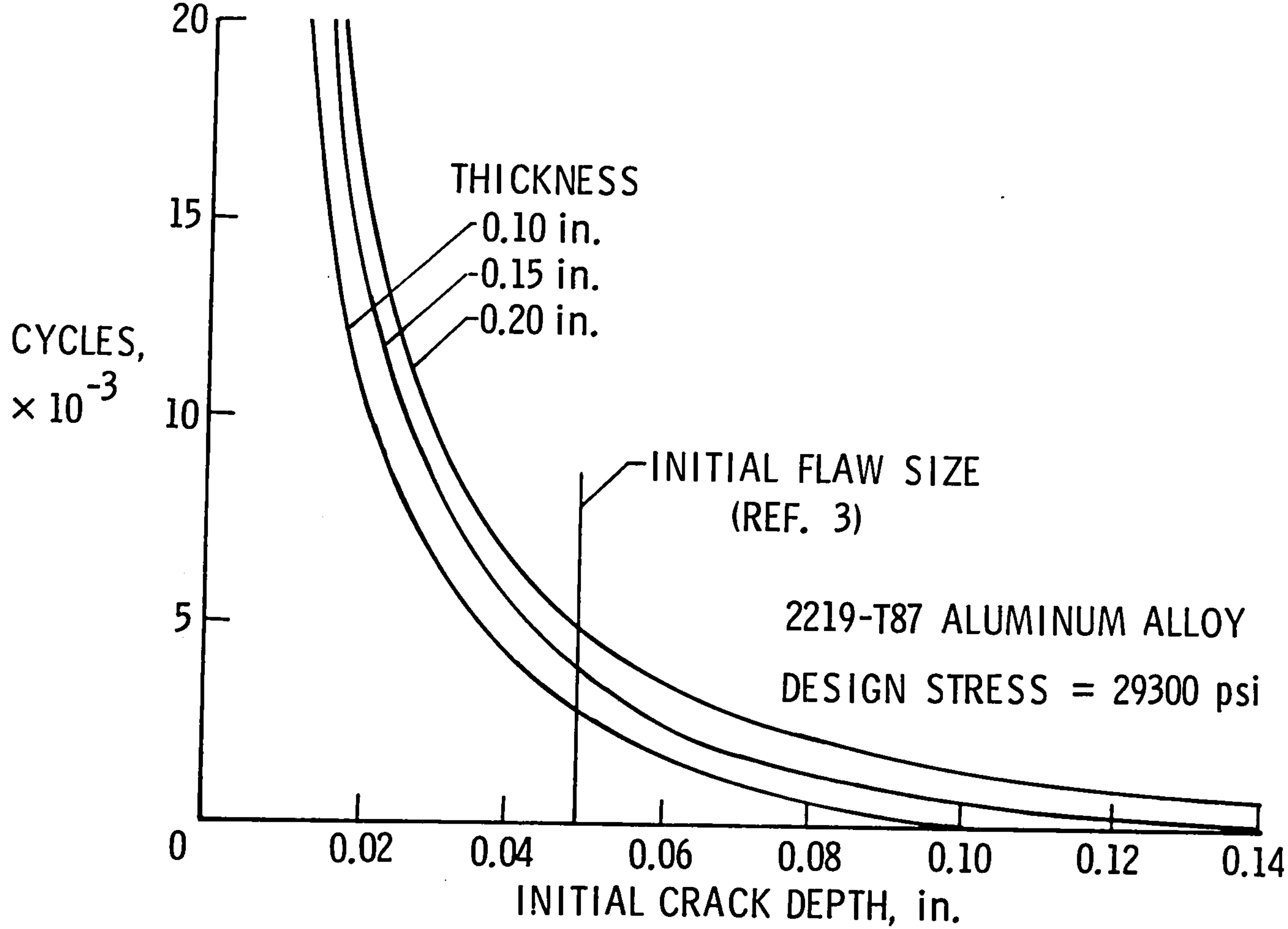

FIG. 5 Tank life versus initial crack depth for three tank wall thicknesses (from Ref. 4).

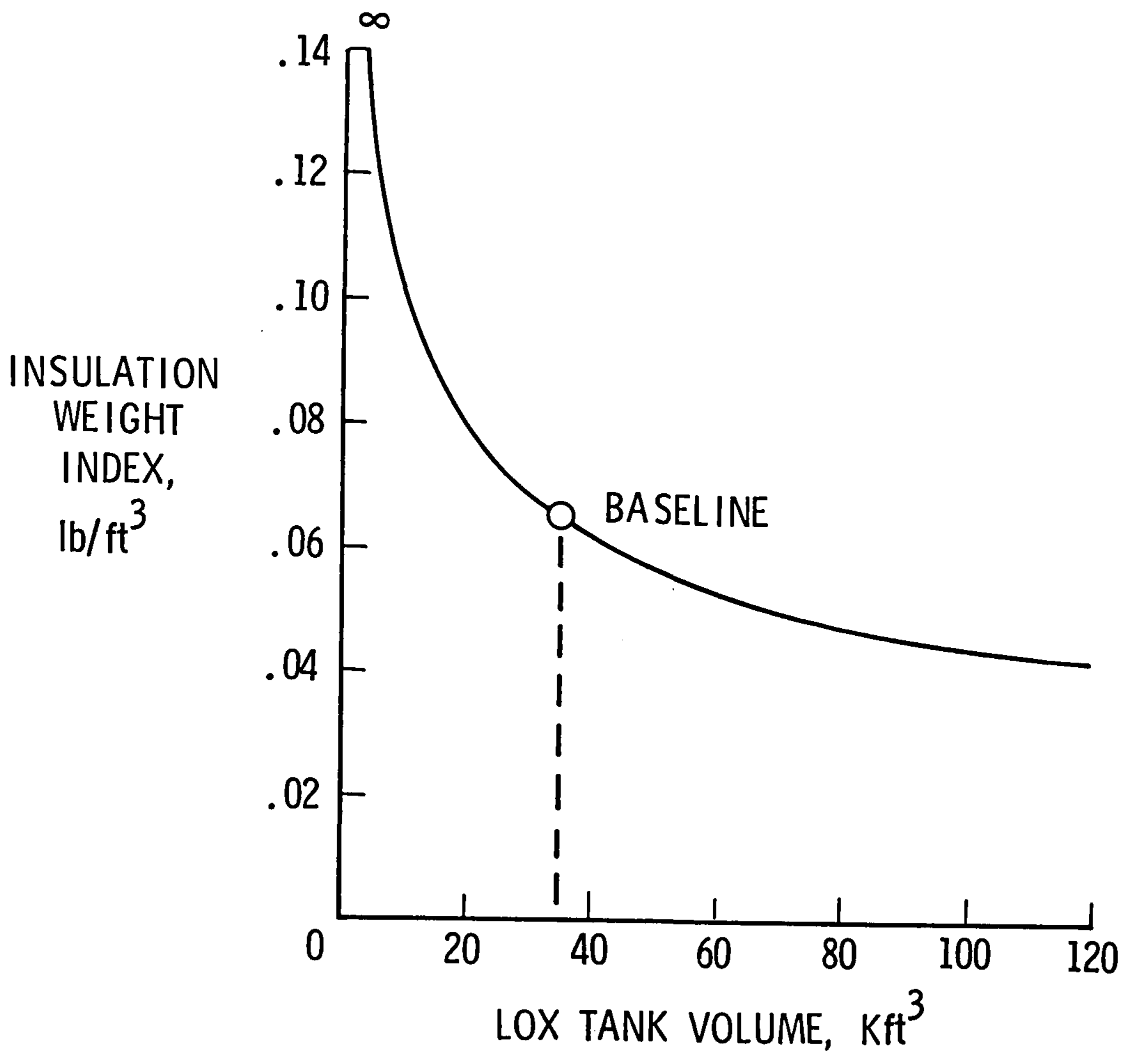

FIG. 6 Insulation weight index versus LOX tank volume.

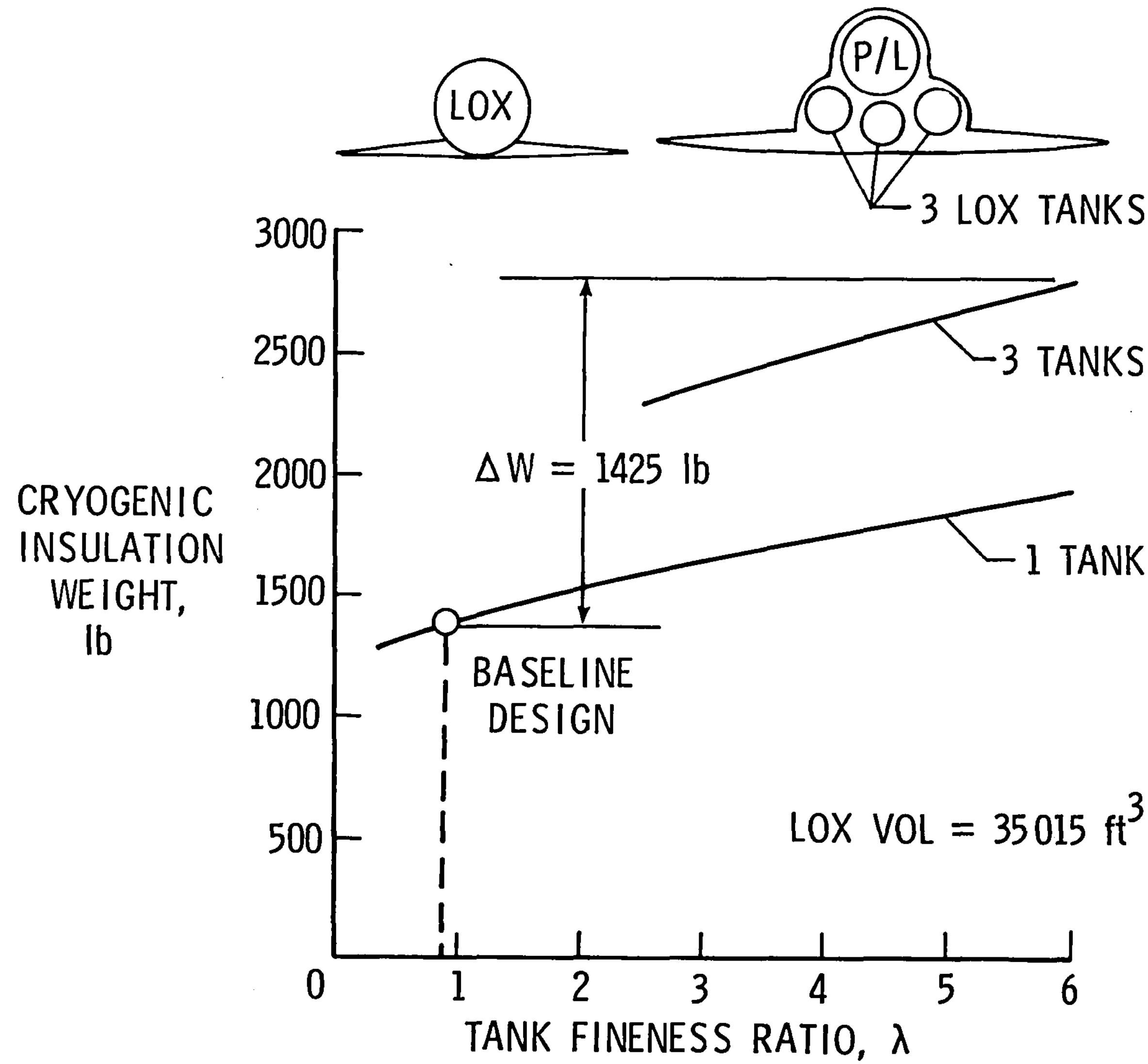

FIG. 7 Cryogenic insulation weight versus LOX tank fineness ratio.

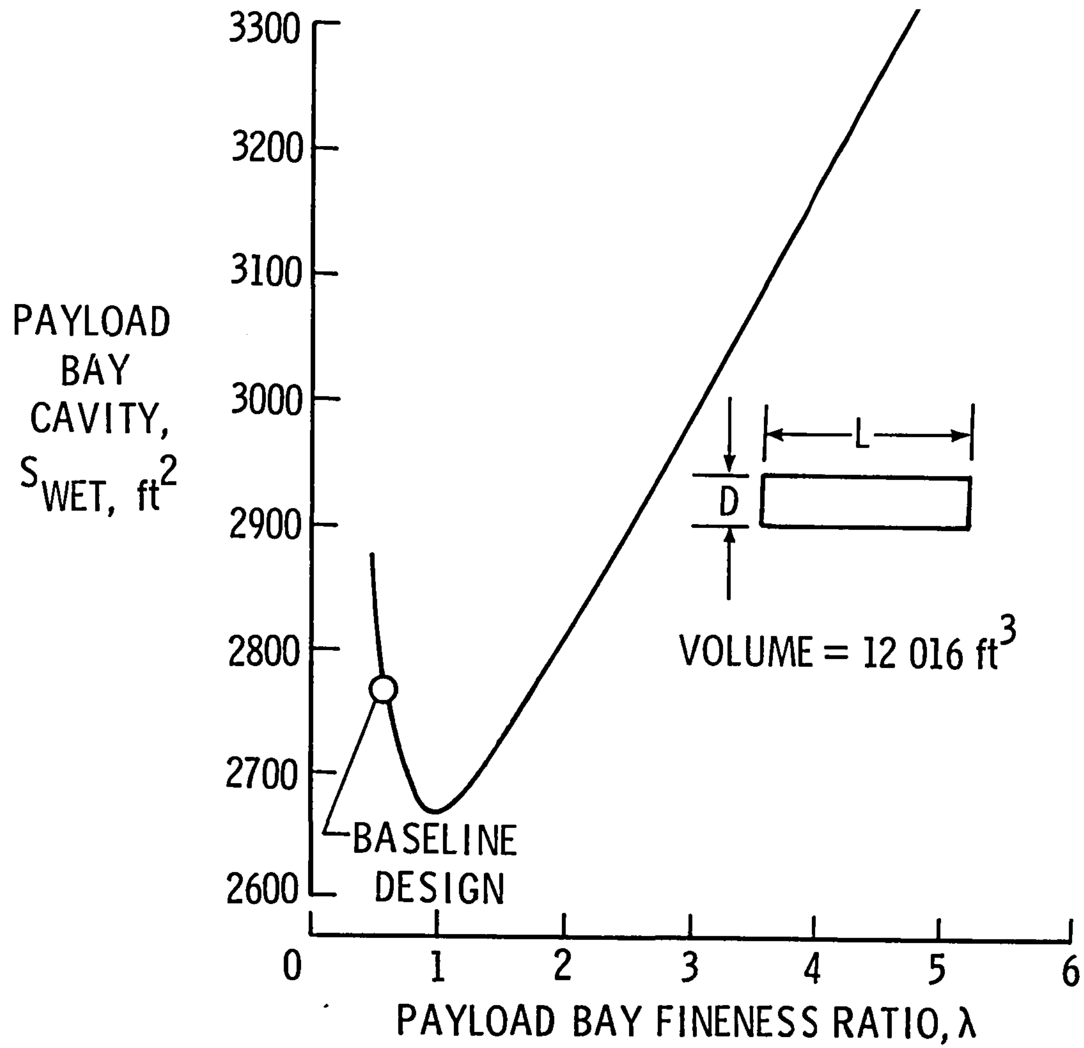

FIG. 8 Payload bay cavity area versus fineness ratio.

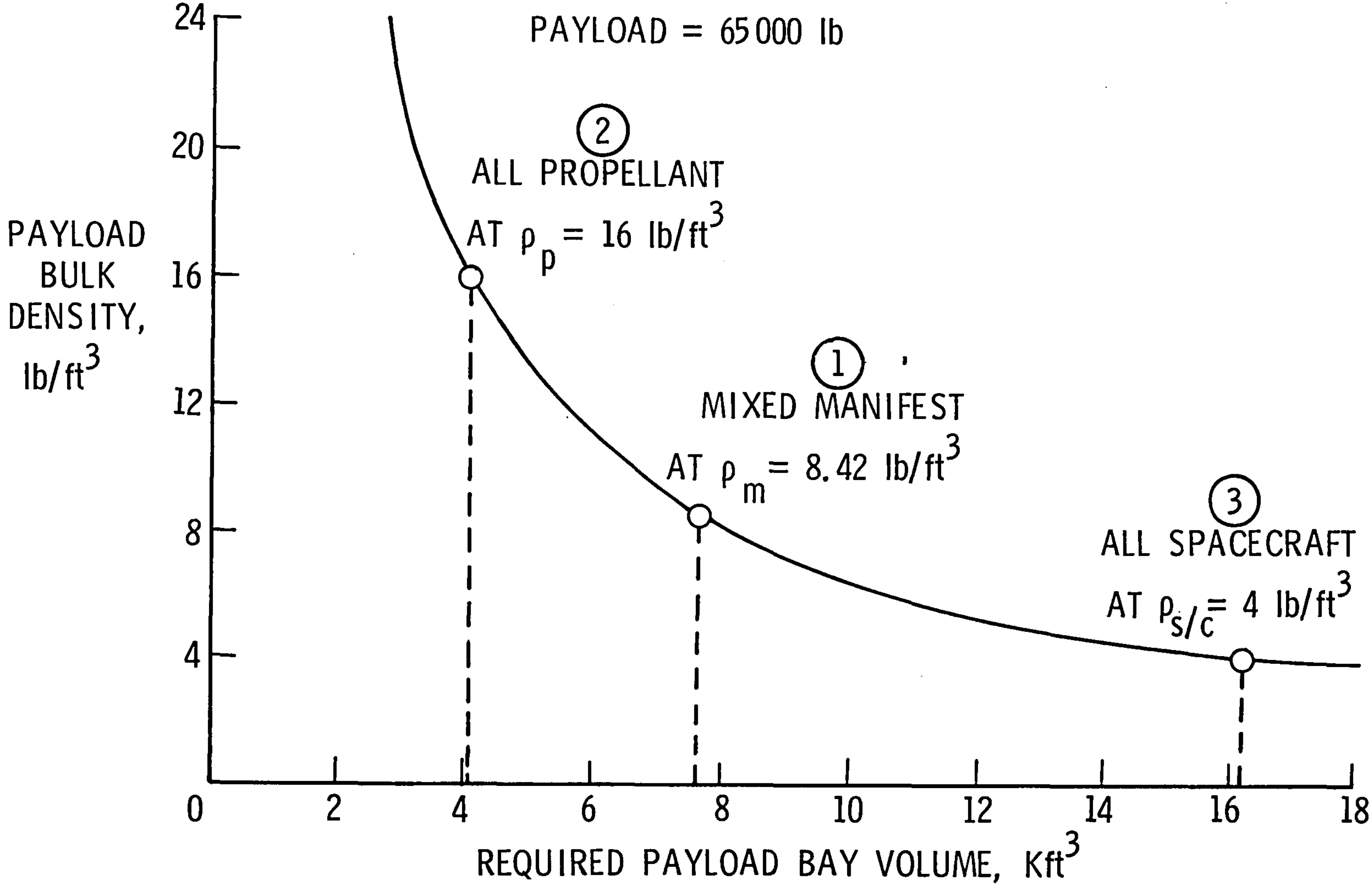

FIG. 9 Payload bay bulk density versus payload bay volume for a 65,000 lb payload.

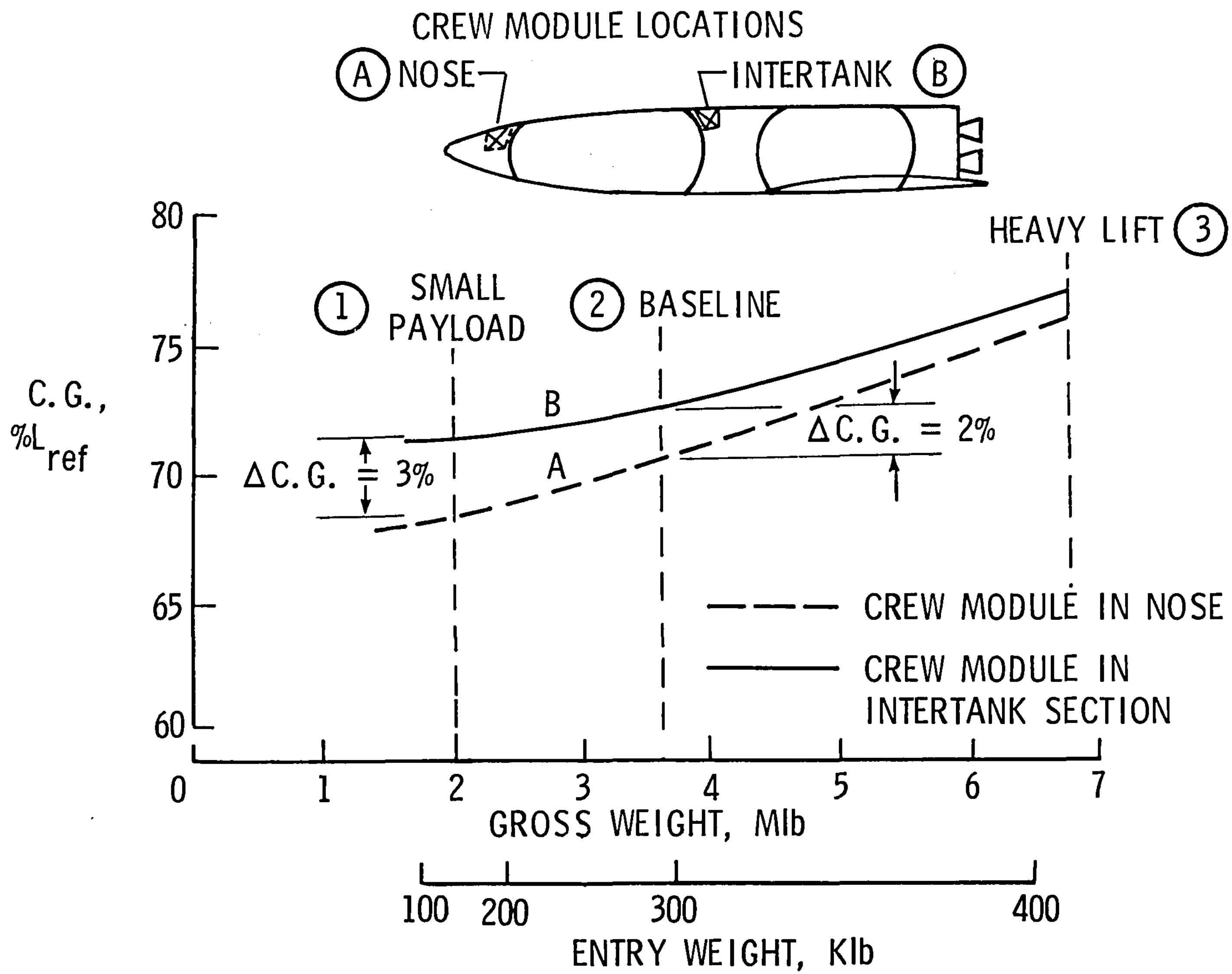

FIG. 10 Impact of crew module location on vehicle entry c.g.

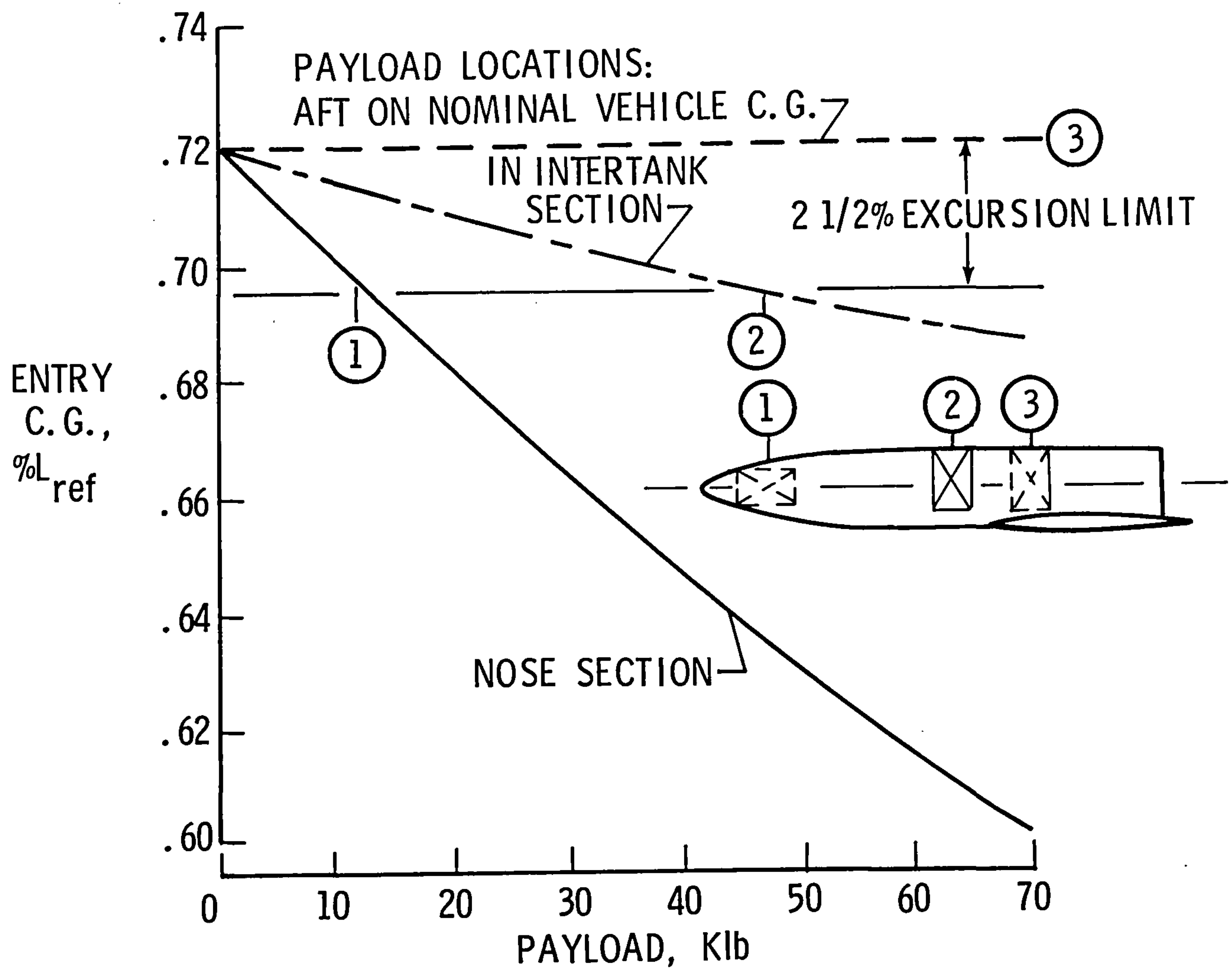

FIG. 11 Vehicle entry c.g. versus payload weight.

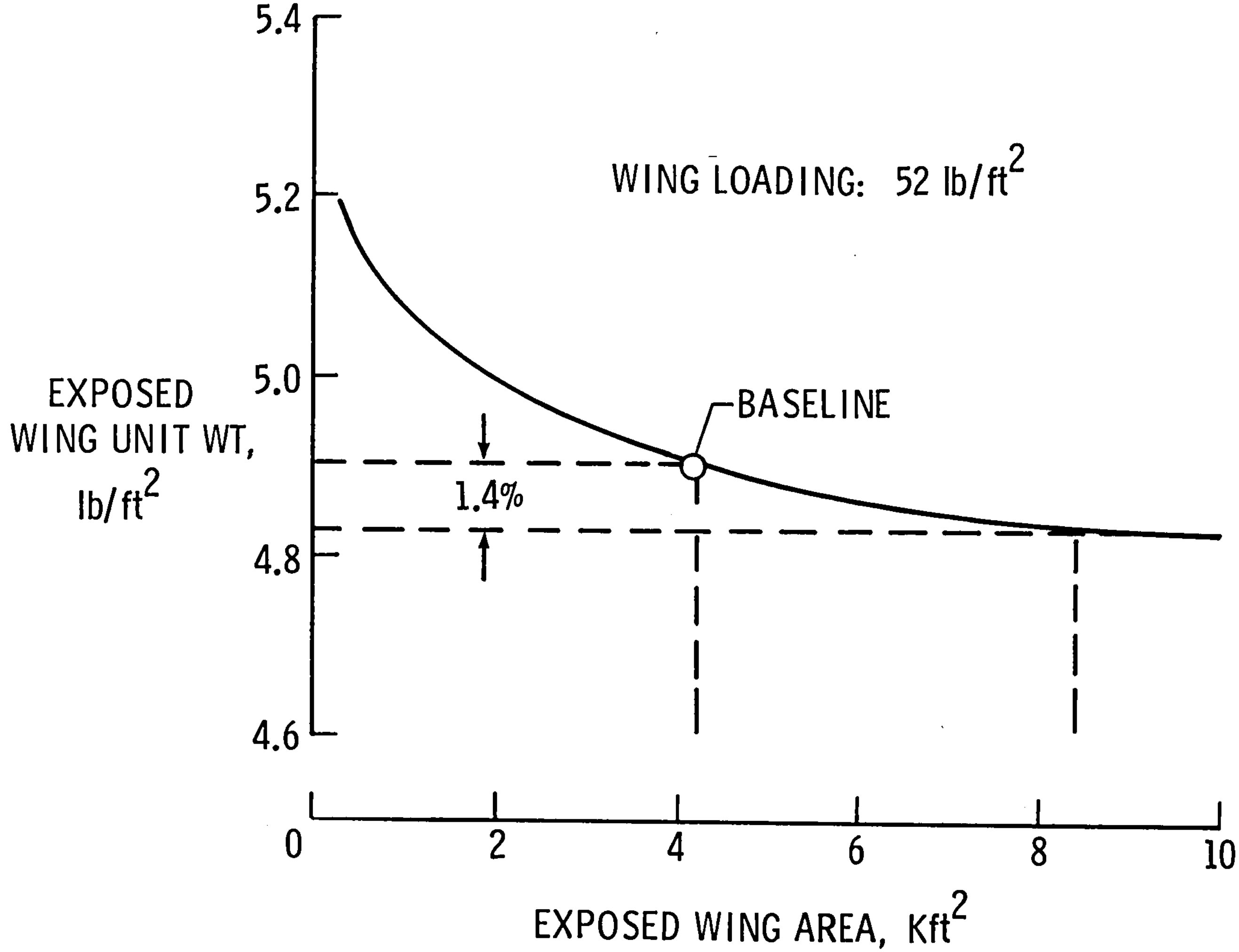

FIG. 12 Exposed wing unit weight versus size.

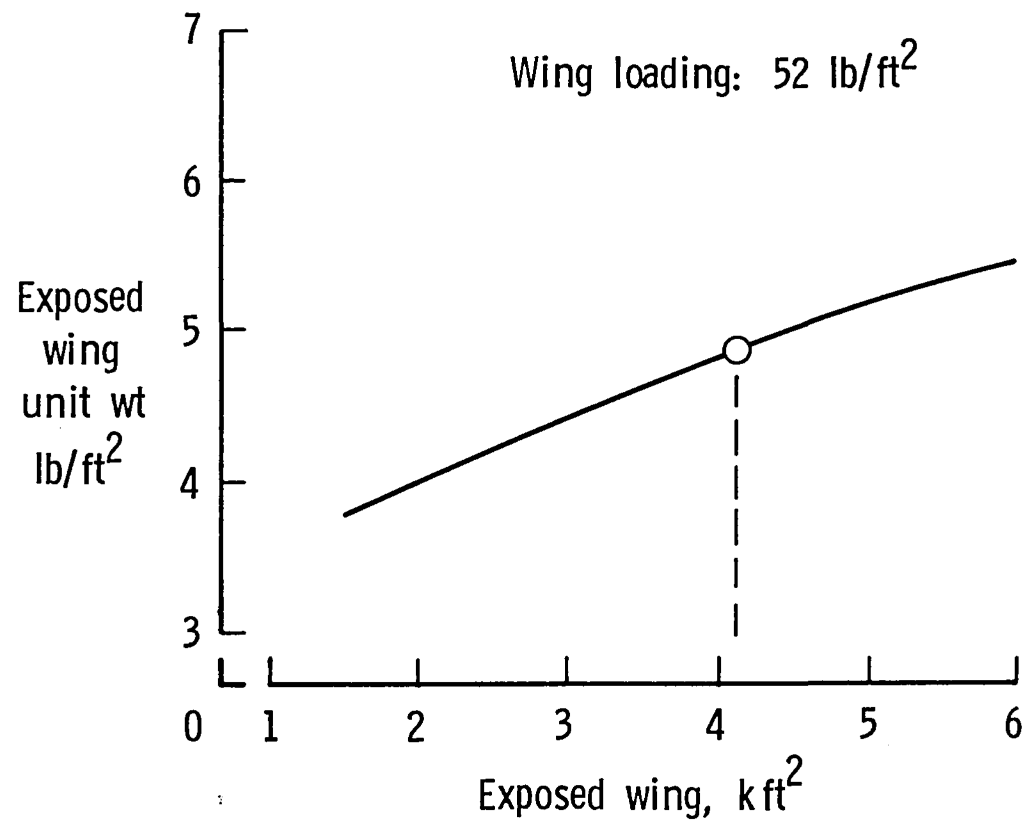

Fig. 12 Exposed wing unit weight versus size.

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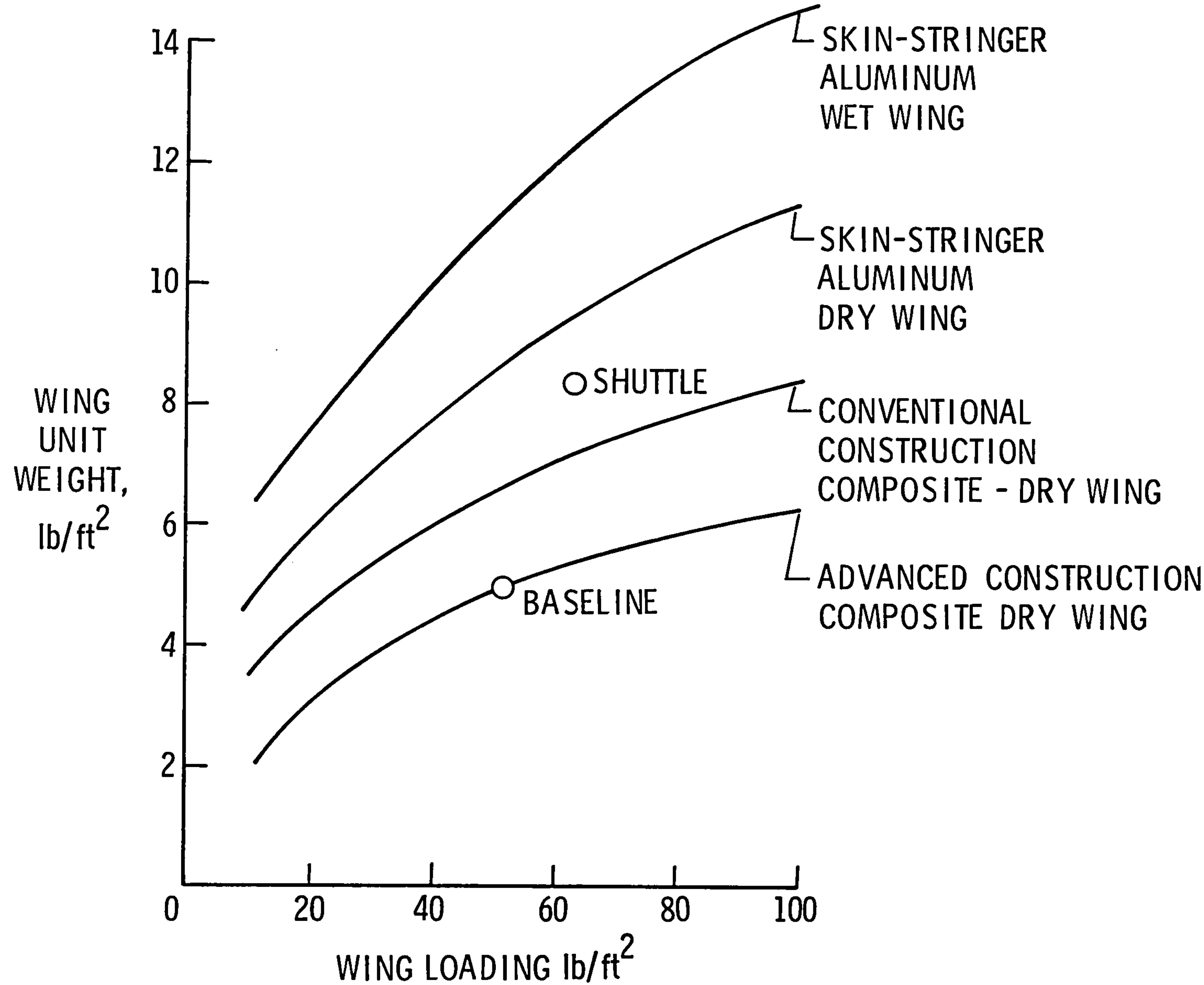

FIG. 13 Exposed wing unit weight versus load for various structural concepts.

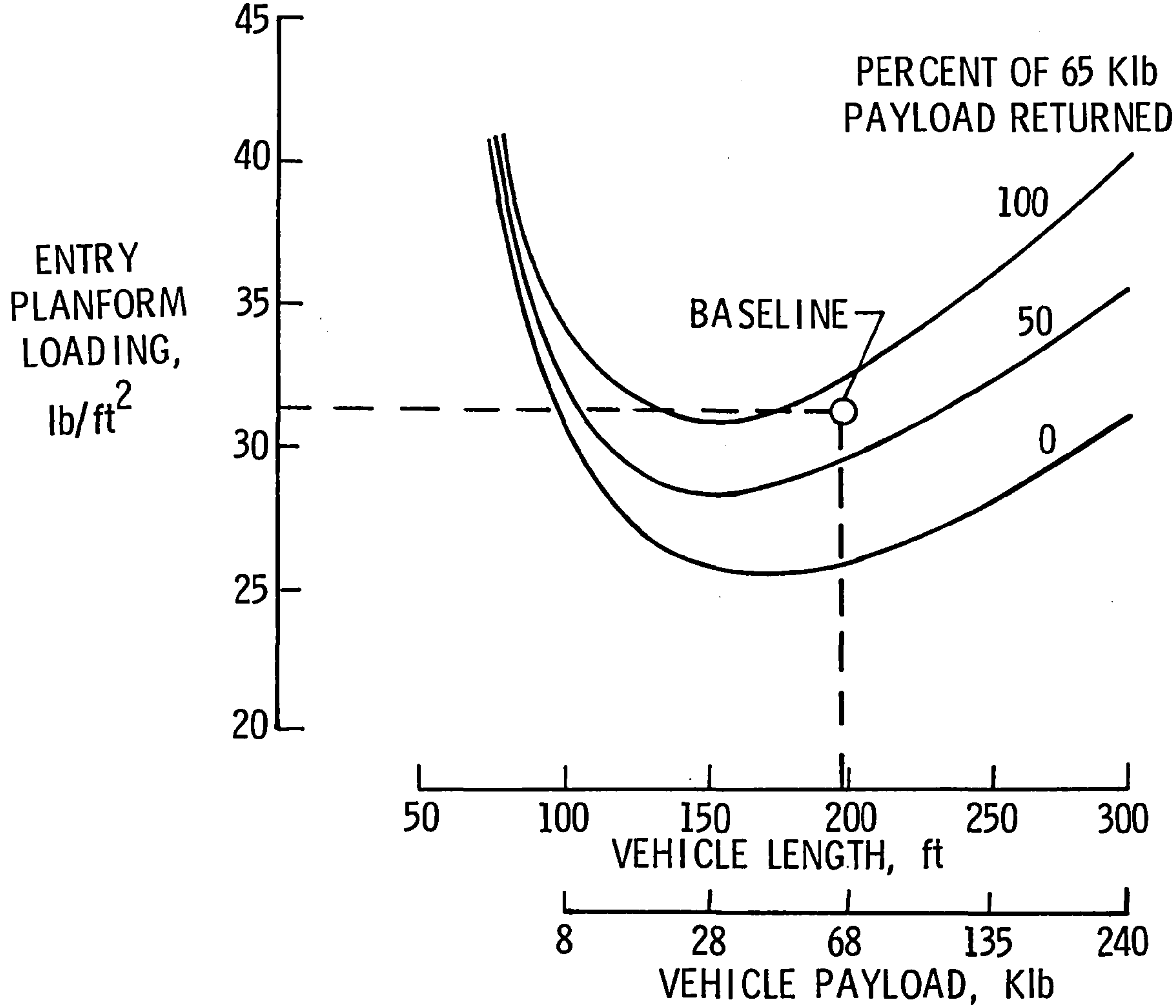

FIG. 14 Entry wing planform loading versus vehicle size for the baseline circular-bodied, single-stage concept.

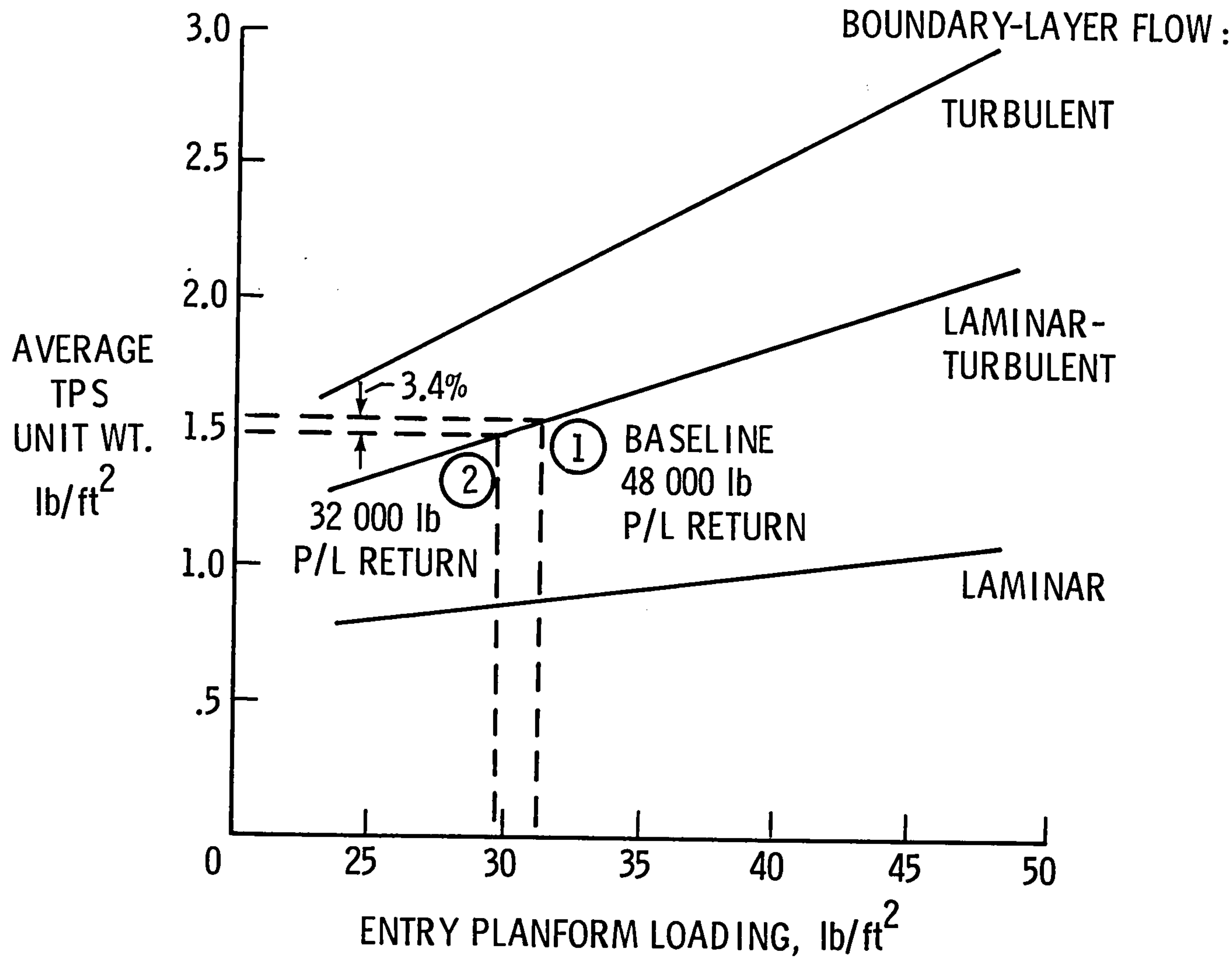

FIG. 15 TPS weight versus entry planform loading.

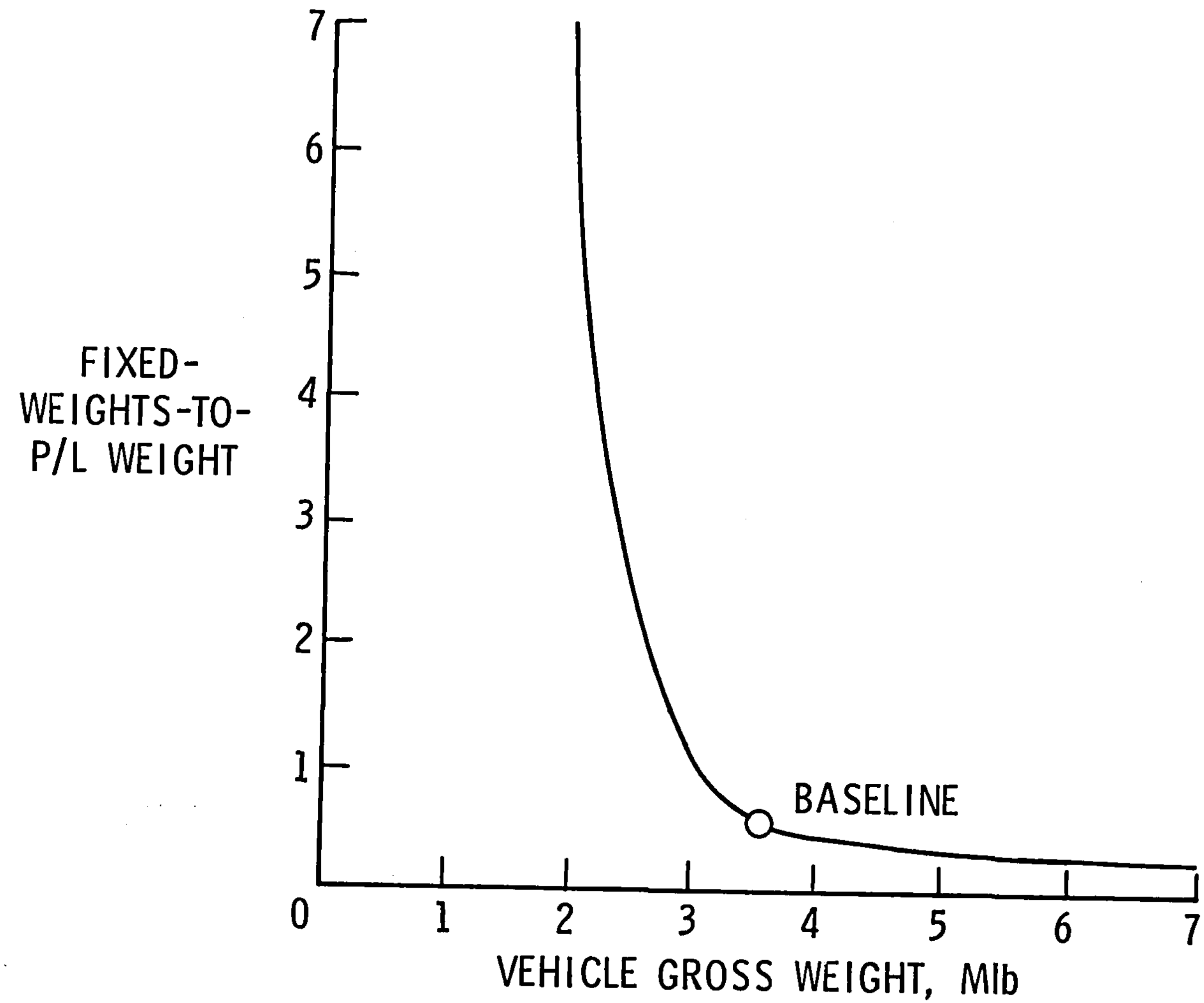

FIG. 16 Fixed weight-to-payload weight versus gross weight for the baseline vehicle.

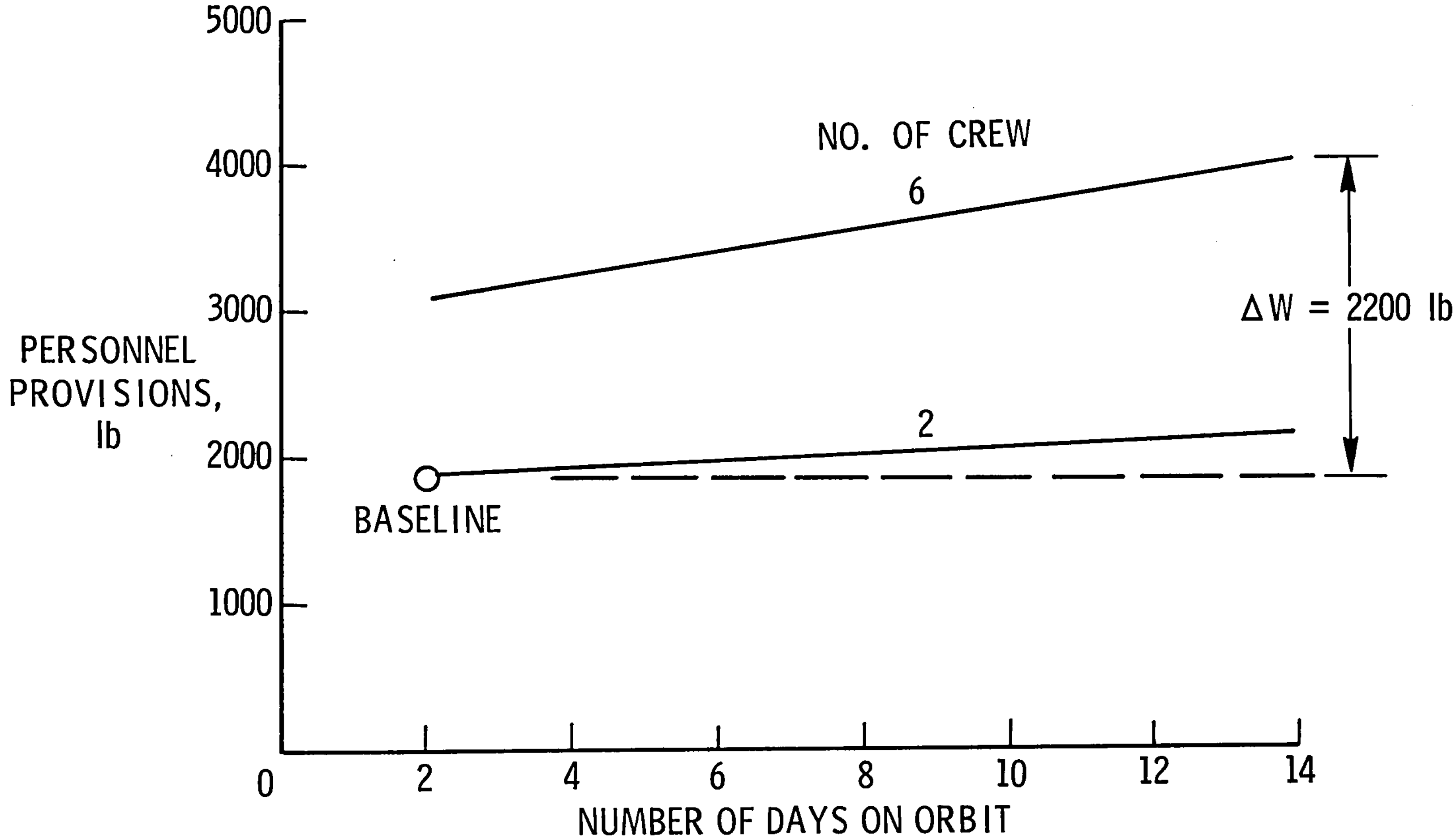

FIG. 17 Personnel provisions versus number of days in orbit.

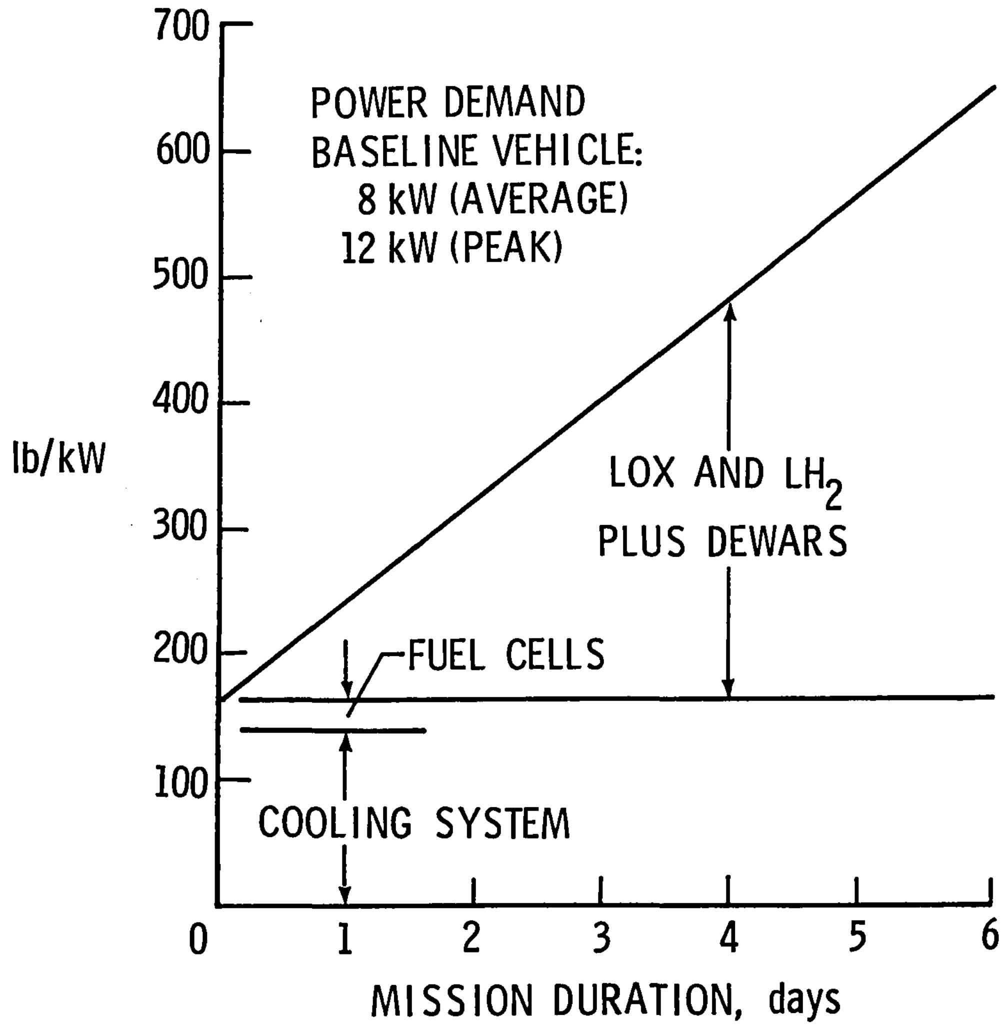

FIG. 18 Power and cooling system weight versus days in orbit.

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15. Supplementary Notes

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16. Abstract

The rate at which subsystem weights grow with vehicle gross weight is assessed and is shown to be critical to the efficiency of large Earth-to-orbit transports. Some subsystems grow as the square of vehicle size, others grow as the cube of vehicle size, and still others remain nearly constant irrespective of vehicle size. The overall trend, however, is a reduction in the inerts as a percentage of gross weight as the vehicle size is increased. For this reason, the larger the vehicle, the greater the payload weight delivered per pound of vehicle manufactured.

Other critical issues addressed include the effects of wing loading and wing size on wing weight, the effect of entry planform loading on thermal protection system weight, the impact of power demand on cooling system and prime power weight, and tank fineness ratio on insulation weight. The effects of body shape and various internal packaging arrangements on weight and balance are also discussed. The greatest impact on overall vehicle weight is body shape and internal packaging, and they could account for weight savings of up to 30 percent in body structure. Other subsystems are important, but the savings are much smaller in relation to overall vehicle weight--individually less than one percent.

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